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LOLA

The Lunar Operations Landing Assembly

A Design Proposal for
the Artemis Program

United States Naval Academy

1 May 1992

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*Extensive collaboration between group members was utilized for each of the design subsystems.

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Chapter 7: Structures .... Gavino Rivas

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CHAPTER 1

INTRODUCTION

MIKE ABREU

Chapter 1 - Introduction

The Lunar Operations Landing Assembly has its origins in the Space Exploration Initiative. In order to effect a manned mission to Mars by the year 2018, it is necessary to return to the Moon and establish a manned presence as a stepping stone to Mars. As the head of the Synthesis Group, LTGEN Thomas P. Stafford, USAF (ret.), said on 16APR92 in a lecture at the U.S. Naval Academy,"...the Moon is a natural lab for science, as well as a platform from which to observe the universe... it can also be a source of materials and energy for further space exploration...". Discussing a timeline for the return to Mars, LTGEN Stafford outlined a manned return to the Moon by 2003-2005, and a manned Mars mission by 2014-2016.

In order to return to the lunar surface by 2005, it is necessary to conduct scientific experiments on the surface of the moon. These experiments would include such topics as detailed lunar topography, substrata mineral and ore content, astronomical observation, and performance of materials on the lunar surface. All of these experiments would be intended to prepare for the manned return to the Moon's surface and the initialization of a lunar colonization program at a specific location. Also, the Moon could be used as a testing ground for an intended Mars surface explorer. In discussing mission parameters and priorities, LTGEN Stafford pointed out a mission implementation hierarchy. The major factors included in this are safety, cost, performance, and schedule.

In an article in *Space News*, Mike Griffin, of the Space

LOLA
Mission Statement

Because the President of the United States has begun the Space Exploration Initiative (SEI), which entails a manned mission to Mars by the year 2016, it is necessary to use the Moon as a stepping stone to this objective. In support of this mission, unmanned scientific exploration of the Moon will help re-establish man's presence there and will serve as a basis for possible lunar colonization, setting the stage for a manned Mars mission.

The lunar landing platform must provide support to its payload in the form of power, communications, and thermal control. The design must be such that cost is held to a minimum, and so that a wide variety of payloads may be used with the lander.

Fig. 1-1 Origin of the LOLA mission. This broad mission statement was obtained from an unclassified article by Mike Griffin at the SEI office.

Exploration Initiative Office, discussed basic requirements and desired specifications for the unmanned lunar lander mission. Objectives and requirements, as well as mission constraints, were obtained from this article, and were the major contributing factors to the LOLA design. Also, the Viking and Surveyor missions, to Mars and the Moon, respectively, were analyzed for possible assistance in concept characterization and mission concept selection.

The mission needs and requirements analysis included many factors crucial to mission feasability and performance. The data and communications bit rate requirements were approximated for anticipated payloads, and the power requirements were also predicted from communications, payload support, and design life requirements. Also, off-the-shelf construction was a major design parameter, which would serve the two-fold purpose of holding down

LOLA Mission Objectives
Primary Objective: **To further the United States' Space Exploration Initiative by returning to the moon with unmanned scientific experiments.**
Secondary Objectives: **To demonstrate to the public that experimental payload missions are feasible.** **To provide a common lunar lander platform so select scientific packages could be targeted to specific lunar locales.** **To enable the lander to be built from off-the-shelf hardware.** **To provide first mission launch by 1996.**

Fig. 1-2 LOLA Mission Objectives.

costs and enabling first launch capability by 1996 by decreasing design maturity time. This would give about nine years of scientific exploration before the intended manned mission.

The LOLA mission also included the secondary objective of demonstrating to the public that in these economically trying times an unmanned mission to the lunar surface is feasible. This is particularly important, since this is the first step in the manned mission to Mars, and cost is of great concern to the American public. If this initial mission is extroaordinarily expensive, it would adversely affect public support and in turn, funding for future SEI programs.

One of the key design obstacles was propulsion type and

REQUIREMENTS AND CONSTRAINTS
- Approx. 440 lbm (200 kg) to lunar surface. - 1000 BPS for communication/data links. - Approx. 135 W total power requirement to lunar surface for > 2 years. - Use of off-the-shelf components to drive down cost. - Provide power for limited experiments through the lunar night. - First mission launch by 1996.

Fig. 1-3 LOLA Design Requirements and Constraints.

requirements to get the specified payload weight to the lunar surface with minimum structure size and mass. A low-weight structure was required, which drove the design towards the use of composite materials and plastics. Another key obstacle was the thermal control of the spacecraft and its payload. The lander platform would have to provide power and thermal control to the payload during both the lunar day and night, each approximately fourteen Earth days in length. In order to effect this, composite materials with wide temperature tolerances were chosen. Also, the platform would have to provide a means for the thermally-enclosed payload to reach the lunar surface directly. To do this, a "shutter" system involving a retracting door was incorporated into the design.

The key elements of design subject to trade were mission operations, where the level of automation can be adjusted, the

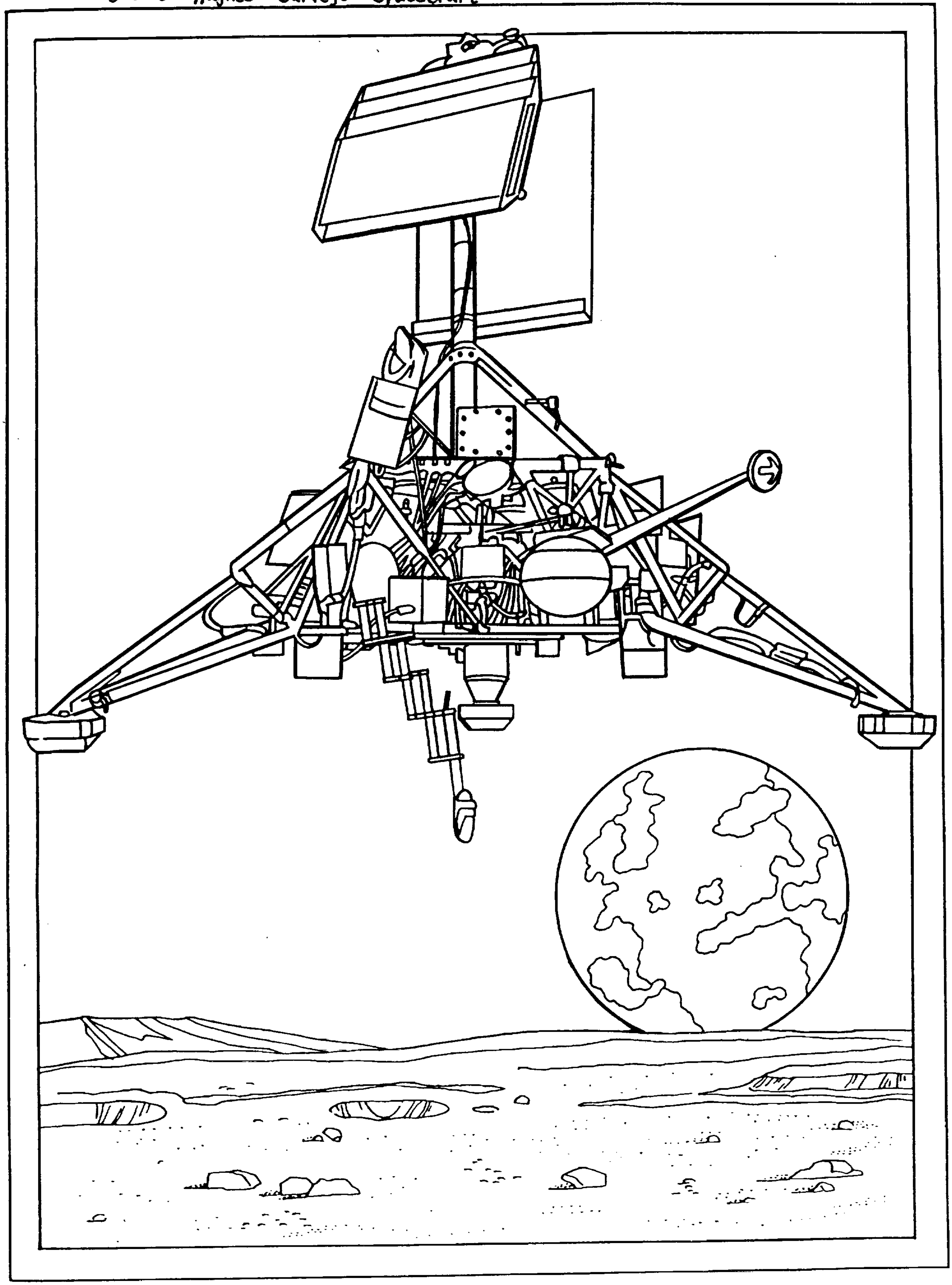
Element	Reason
Launch System	Can choose minimum cost for desired orbit and s/c sizing
Orbit	Options are geostat. transfer orbit or direct path to surface, with varying individual mission reqs.
Spacecraft Bus	Multiple options based on comms. and data rate, power
Mission Operations	Level of automation can be adjusted, during transit or on surface

Fig. 1-4 Elements Subject to Trade for LOLA.

launch system, which depends on cost and size of the lander for a particular orbit, the spacecraft bus, and the orbit desired. Together, these options subject to trade enable great flexibility in design and in overall mission parameters for future missions as the design matures. Problems and tradeoffs for each subsystem are included in each chapter, as well as component selection criteria.

All of these mission requirements, constraints, and drivers combined to create a low-cost spacecraft that will place a scientific payload on the lunar surface by 1996. In doing so, the first step on the road to Mars will have been taken, and it is absolutely imperative that this design prove the feasability and effectiveness of unmanned lunar exploration.

Fig. 1-5 Hughes' Surveyor Spacecraft



CHAPTER 2

PROPULSION AND ORBITS

FERNANDO ARGELES

Chapter 2 - Propulsion

Requirements:

The overriding requirement of this system is to accomplish the mission of landing 200 kg of payload on the surface of the moon.

Launch system:

The Delta II rocket is the launch vehicle because it is the smallest vehicle that can complete the mission. The payload faring is 2.9 m in diameter and 3 m tall. The Delta II can lift 3850 kg of mass into low earth orbit (LEO about 400 km altitude).

Orbit calculations:

In order to get LOLA to the moon, Hohmann transfers were utilized. The transfer ellipse has its perigee at earth's LEO and the apogee at the orbit of the moon. The patched conic approximation was used to determine the velocity of LOLA at the moon. In the year of 1996 when the LOLA is first scheduled to launch the plane of the moon's orbit will coincide with the parking orbit of LOLA. For this reason a plane change will not be necessary to reach the moon. This means that the launch in 1996 would be the lowest energy launch window for transit to the moon. Figures I and II at the end of this chapter have the diagrams for the patched conic approximation of LOLA's transit to the Moon.

Propulsion system used:

The fundamental reason for the use of liquid rockets instead of solid rockets to provide the velocity needed at low earth orbit was two fold. First the liquid rockets can be controlled after firing. Should difficulties be encountered at the burn to send LOLA in the transfer ellipse it will not result in a total loss of the vehicle. Second the use of liquid rockets allows for better space management in the Delta II payload faring. It is thus possible to give the payload the a larger volume.

Figure 3 at the end of the chapter has the schematic for the Earth - Moon stage of the propulsion system. The same schematic for the landing stage is used however there are three engines in the engine assembly. Once the burn to get LOLA out of LEO has been completed the fuel tank assembly will be discarded and only the lander will continue onto the moon.

The following pages are the equations that were used to determine the fuel required and the orbital trajectory. All the calculations were done on computer using the spreadsheet Supercalc 4.

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EARTH - MOON DELTA V CALCULATIONS

Distance from earth to moon = Rem = 384400 km
Distance from earth center = Rp = 6778 km
low earth orbit (parking) = 400 km alt
semi major axis for hohmann transfer = a = 198778 km

Velocity in low earth orbit (400 km altitude)

$$V(leo) = \sqrt{\frac{\mu_e}{R}}$$

μ_e = 398600 km^3/sec^2
R = 6378 km + 400 km

$$V\ (leo) = \sqrt{\frac{398600\ km^3/sec^2}{6778\ km}}$$

V (leo) = 7.669 km/s

Velocity needed at perigee of transfer ellipse

$$Vp = \sqrt{\mu_e\left(\frac{2}{R} - \frac{1}{a}\right)}$$

$$= \sqrt{398600\ km^3/sec^2 \left(\frac{2}{6778\ km} - \frac{1}{198778\ km}\right)}$$

Vp = 10.752 km/s

Delta V required to get into the transfer ellipse to the moon.

Delta V = Vp - V (leo)

= 10.752 km/s - 7.669 km/s

= 3.083 km/s

Delta V calculations for the burns required to stop at the surface of the moon are explained in the patch conic section.

PATCH CONIC VARIABLES

D = distance from earth to the moon

gamma 1 = γ_1 = angle from moon to earth center to patch point

lambda 1 = λ_1 = angle from patch point to moon center to center of earth

r_1 = distance from earth center to patch point

r_0 = distance from earth center to LOLA at LEO just before thruster firing

v_0 = velocity of LOLA after firing thruster to get to the moon

theta = θ = angle from patch point to earth center to location of moon at time of thruster firing in LEO

r_2 = r_s = radius of moon's sphere of influence

v_2 = LOLA's velocity with respect to moon after patch is complete

v_1 = LOLA's velocity at patch point with respect to the earth

epsilon 2 = ϵ_2 = the angle between the velocity of LOLA after patch and center of moon

h = angular momemtun

p = semi latus rectum

rp = radius of periselenium

vp = velocity at periselenium

Values of variable at the patch conic section point

λ_1 = 1.462 rad	h = -4308.78 km^2/s
r_1 = 382896.8 km	p = 3786.7 km
v_1 = .2678 km/s	energy = .2562
ϕ_1 = .7802 rad	e = 1.1814
γ_1 = .1729 rad	rp = 1735 km
v_2 = .8126 km/s	vp = 2.480 km/s
ϵ_2 = -.0801 rad	r_2 = 66300 km

EQUATIONS FOR PATCHED CONIC CALCULATIONS

Earth - Moon trajectory before Patch point

$$\text{Energy} = \frac{Vo^2}{2} - \frac{\mu_e}{Ro}$$

$$= \frac{(10.752\ \text{km/s})^2}{2} - \frac{398600\ \text{km}^3/\text{sec}^2}{6778\ \text{km}}$$

$$\text{Energy} = -1.00516\ \text{km}^2/\text{sec}^2$$

$$h = \text{angular momentum} = r_o v_o \cos(\phi_o)$$

$$= (6778\ \text{km}) * (10.752\ \text{km/s}) * \cos(90)$$

$$h = 72977.056\ \text{km}^2/\text{sec}$$

$$r_1 = \sqrt{D^2 + Rs^2 - 2 * D * Rs * \cos\lambda_1}$$

$D = 384,400$ km

$Rs = 66,300$ km

$$v_1 = \sqrt{2\ (\text{energy} + \mu_e/r_1)}$$

$\mu_e = 398600\ \text{km}^3/\text{sec}^2$

$$\phi_1 = \cos^{-1}\left(\frac{h}{r_1 * v_1}\right)$$

$$\gamma_1 = \sin^{-1}\left(\frac{Rs * \sin\lambda_1}{r_1}\right)$$

$$v_2 = \sqrt{v_1^2 + Vm^2 - 2 * v_1 * Vm * \cos(\phi_1 - \gamma_1)}$$

$Vm = 1.018$ km/s

$$\epsilon_2 = \sin^{-1}\left(\frac{Vm \cos\lambda_1}{v_2} - \frac{v_1 \cos(\lambda_1 + \gamma_1 - \phi_1)}{v_2}\right)$$

EQUATIONS AT PATCH POINT

These are used to determine velocity of LOLA with respect to the moon once it has entered the moons sphere of influence.

All the following terms refer to LOLA in trajectory about the moon.

$$\text{energy} = \left(\frac{V_2^2}{2} - \frac{\mu_m}{R_2} \right)$$

$$h = r_2 * v_2 * \sin \epsilon_2$$

r_2 = 66300 km
v_2 = comes from previous page

$$p = \frac{h^2}{\mu_m}$$

μ_m = 4902.83 km^3/sec^2

$$e = \sqrt{1 + \frac{2 * \text{energy} * h^2}{\mu_m^2}}$$

$$rp = \frac{p}{1 + e}$$

$$vp = \sqrt{2 * (\text{energy} + \mu_m / rp)}$$

This velocity (vp) is the speed at which LOLA would impact the surface of the moon without any change in velocity. By adding the same velocity in the opposite direction during the transit from the patch point to the surface of the moon a soft landing will be achieved at the moon's surface. The delta V required for the breaking at the moon therefore is just the (vp) that is calculated in this section.

FUEL MASS CALCULATIONS

Earth - Moon Trajectory

$$\Delta V = g * Isp * \ln\left(\frac{M_o}{(M_o - M_f)}\right)$$

$M_o = 3492.75$ kg

$g = 9.81$ m/s^2

Isp = 340 sec

$\Delta V = 3083$ m/s

$$3083 \text{ m/s} = (9.81 \text{ m/s}^2) * (340 \text{ sec}) * \ln\left(\frac{3492.75 \text{ kg}}{(3492.75 \text{ kg} - M_f)}\right)$$

$$.92433 = \ln\left(\frac{3492.75 \text{ kg}}{(3492.75 \text{ kg} - M_f)}\right)$$

$$2.52017 = \frac{3492.75 \text{ kg}}{(3492.75 \text{ kg} - M_f)}$$

$$8802.328 \text{ kg} - 2.52017\ M_f = 3492.75 \text{ kg}$$

$M_f = 2106.8$ kg

Lunar landing stage

$$\Delta V = g * Isp * \ln\left(\frac{M_o}{(M_o - M_f)}\right)$$

$\angle V = 2480$ m/s

g , Isp = same as above

$M_o = 913$ kg

$$2480 \text{ m/s} = (9.81 \text{ m/s}^2) * (340 \text{ sec}) * \ln\left(\frac{913 \text{ kg}}{913 \text{ kg} - M_f}\right)$$

$$.74354 = \ln\left(\frac{913 \text{ kg}}{913 \text{ kg} - M_f}\right)$$

$$2.10337 = \frac{913 \text{ kg}}{913 \text{ kg} - M_f}$$

$$1920.376 \text{ kg} - 2.10337\ M_f = 913 \text{ kg}$$

$M_f = 479$ kg

FUEL TANK SIZE CALCULATIONS

Assume equal volumes for the amounts of the liquid bi-propellant.

Fuel used = MMH and N_2O_4

Density of fuel

MMH = .86 g/cm^3

N_2O_4 = 1.43 g/cm^3

Earth - Moon stage

Fuel mass = 2107 kg

volume m^3

MMH = .92008

N_2O_4 = .92008

Lunar Lander stage

Fuel Mass = 479 kg

volume m^3

MMH = .20917

N_2O_4 = .20917

Spherical tanks were used for the storage of both stages of the bi-propellant. In both cases four spheres were used.

volume of a sphere

$$v = \frac{4 * pi * r^3}{3}$$

Since the volume is known for the fuel mass the equation for determining the radius of the respective spheres is:

$$r = \sqrt[3]{\frac{3\ v}{4 * pi}}$$

There are four spheres carrying the fuel thus the volume that each tank has to carry is only half that stated above.

$$r = \sqrt[3]{\frac{3\ v}{8 * pi}}$$

Radius of tanks

Earth - Moon stage = .479 m

Lunar Lander stage = .292 m

TRANSIT TIME CALCULATIONS

Period of transfer orbit

T = period
a = semi major axis = 198778 km
= 398600 km^3/sec^2

$$T = 2 * pi * \left(\frac{a^3}{\mu_e}\right)^{1/2}$$

$$T = 2 * pi * \left(\frac{(198778\ km)^3}{398600\ km^3/sec^2}\right)^{1/2}$$

T = 881990 sec

Transit time to the moon is half the orbital period for the transfer ellipse.

Transit time = P/2
= 881990 sec / 2
= 440995 sec
= 7349.9 min
= 122.448 hours

= 5.104 days

Orbital period of Moon about earth

$$T = 2 * pi * \left(\frac{a^3}{\mu_e}\right)^{1/2}$$

$$T = 2 * pi * \left(\frac{(384400\ km)^3}{398600\ km^3/sec^2}\right)^{1/2}$$

T = 2371844.9 sec

Part of orbit moon will complete during LOLA's transit time to the moon.

$$\text{part of orbit} = \frac{440995\ sec}{2371844.9\ sec}$$

= .1859 rev
= 1.1682 rad

= 66.93 degrees

It is thus necessary to launch LOLA 66.93 degrees ahead of the moon so that the two orbits will cross at the same time. LOLA will thus reach the apogee of the transfer orbit when the moon is at the same point.

THRUSTER FIRING CALCULATIONS

It was assumed in the orbital calculations that all the burns would be instant. It however is known that the actual thruster firing times will not be infinitely short. This section is the calculation of the thruster firing time taking into account the amount of fuel and thrust developed in the engines.

Earth - Moon trajectory

Engine used
XLR - 132 (developed by Rocketdyne)

thrust = 16700 N
g = 9.81 m/s^2
Isp = 340 sec
Mass of fuel = 2107 kg

$$\dot{m} = \frac{F}{Isp * g}$$

$$\dot{m} = \frac{16700 \text{ N}}{340 \text{ sec} * 9.81 \text{ m/s}^2}$$ note N = kg m/s^2 *m

$$\dot{m} = 5.0069 \text{ kg/s}$$

Thruster firing time = t

$$t = \frac{2107 \text{ kg}}{5.0069 \text{ kg/s}}$$

$$t = 420.82 \text{ sec}$$
$$= 7.01 \text{ min}$$

Lunar Lander stage

engines used
HS 601 - developed by ARC/LPG

thrust = 489 N per engine Isp = 340 sec
Mass of fuel = 479 kg

$$m = \frac{F}{Isp * g}$$

$$m = \frac{489 \text{ N} * 3}{340 \text{ sec} * 9.81 \text{ m/s}^2}$$

$$m = .43983 \text{ kg/s}$$

Thruster firing time = t

$$t = \frac{479\ \text{kg}}{.43983\ \text{kg/s}}$$

$$t = 1089.1\ \text{sec}$$
$$= 18.15\ \text{min}$$

It should be noted that these engines will not only be used to slow LOLA down when it arrives at the moon but also to stabilize the landing. A computer will control the engine firing and will take the lander all the way down to the surface of the moon with a 0 m/s landing speed.

SUMMARY OF PROPULSION AND FINAL WEIGHT TABLE

Launch vehicle = Delta II

Earth - Moon stage

delta v required = 3083 m/s
rocket used = XLR - 132 (Rocketdyne)
fuel used = MMH and N_2O_4

mass of fuel	= 2107 kg
mass of rocket	= 51.26 kg
mass of tanks and support	= 421.4 kg

Lunar landing stage

delta v required = 2480 m/s
rockets used = HS 601 (ARC/LPG)
fuel used = MMH and N_2O_4

mass of fuel	= 479 kg
mass of rocket (3)	= 12.24 kg
mass of tanks and support	= 71.85 kg

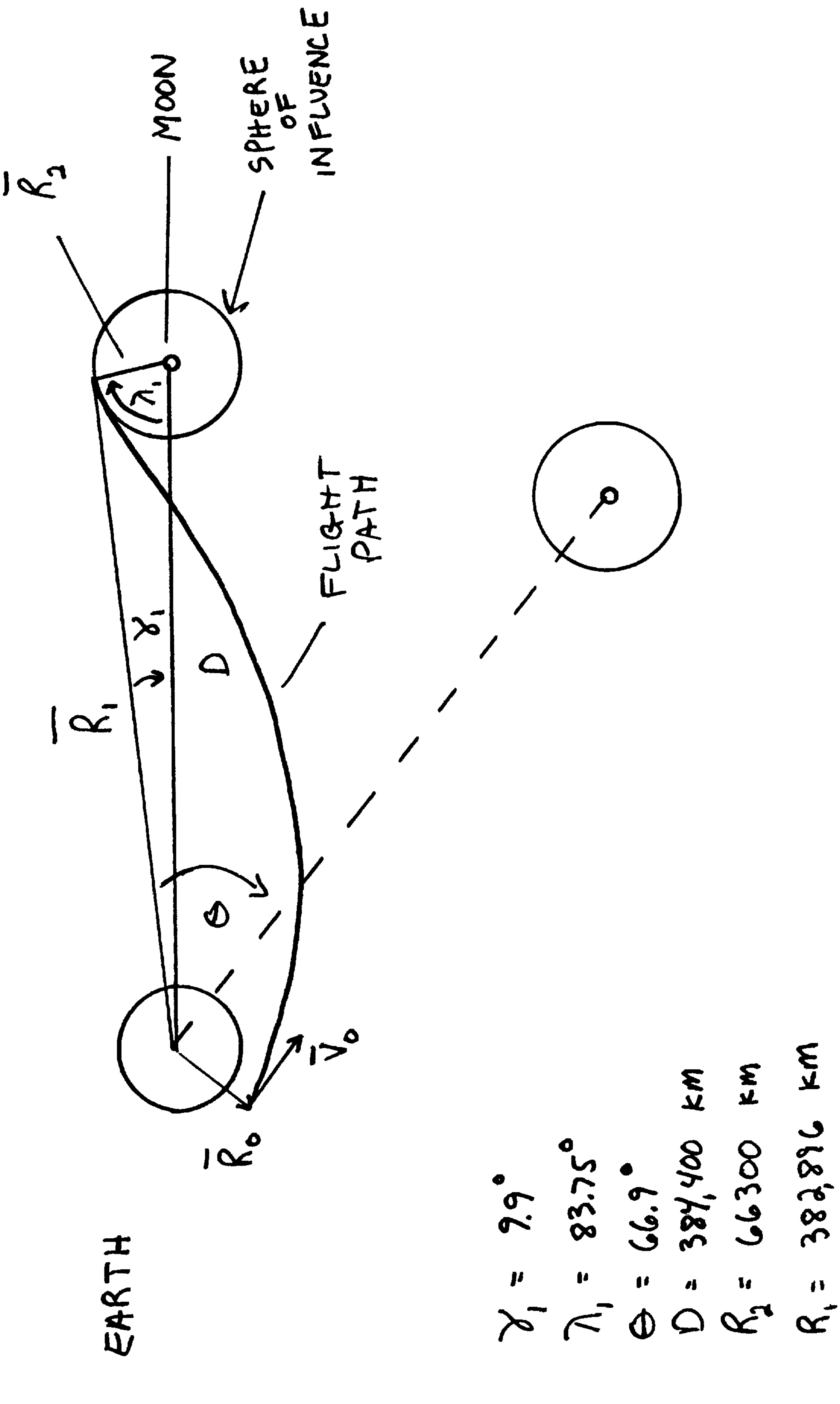
Attitude control

mass of fuel	= 2 kg
mass of thrusters	= 1.2 kg

Total mass for the propulsion system = 3145.95 kg

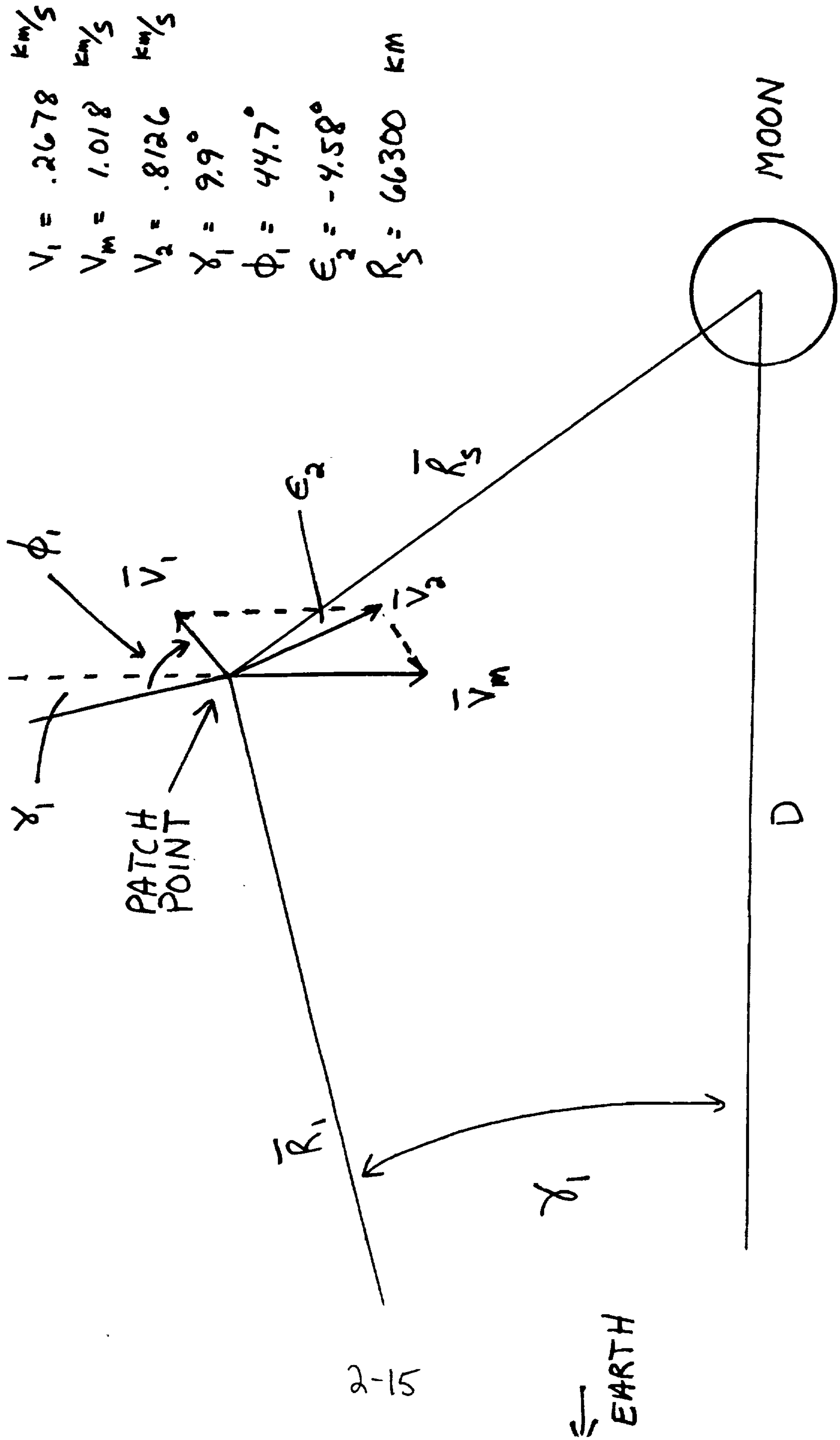
The reason for the large amount of mass required for the fuel tanks and support is that a gas blowdown system is used to feed the fuel to the rockets. Both the Earth - Moon and the lunar lander stages use this assembly and it requires substantial mass. The gas in the fuel blowdown system is N_2.

FIGURE 1



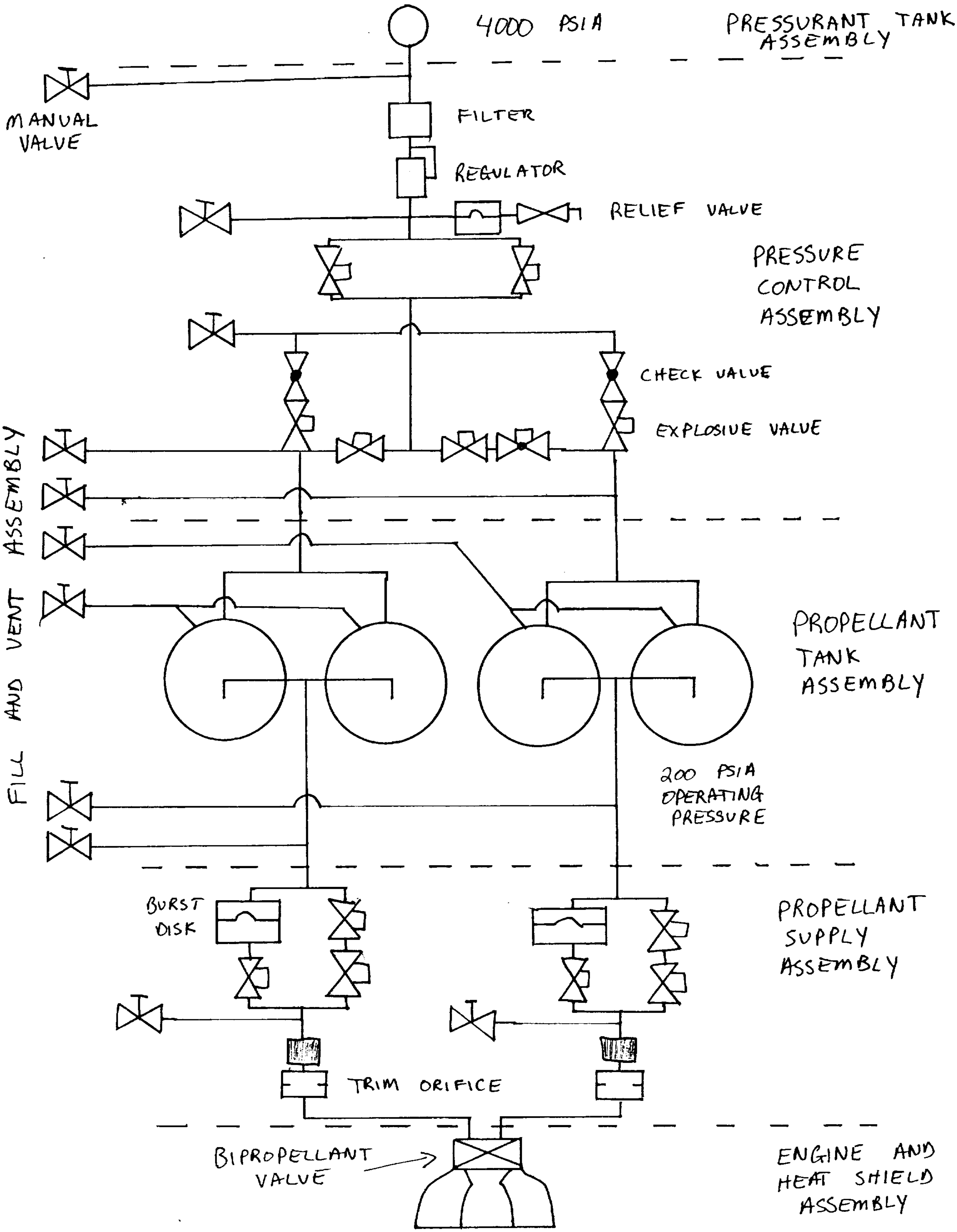
MOON
SPHERE OF INFLUENCE
FLIGHT PATH
EARTH
$\bar{R}_2$
$\bar{R}_1$
λ_1
γ_1
D
θ
$\bar{V}_0$
$\bar{R}_0$
$\gamma_1 = 9.9^\circ$
$\lambda_1 = 83.75^\circ$
$\theta = 66.9^\circ$
D = 384,400 KM
R_2 = 66300 KM
R_1 = 382,896 KM

FIGURE 2



$V_1 = .2678$ km/s
$V_m = 1.018$ km/s
$V_2 = .8126$ km/s
$\gamma_1 = 9.9°$
$\phi_1 = 44.7°$
$\epsilon_2 = -4.58°$
$R_s = 66300$ km
MOON
$\bar{R}_s$
ϵ_2
ϕ_1
$\bar{V}_1$
$\bar{V}_2$
$\bar{V}_m$
γ_1
PATCH POINT
D
$\bar{R}_1$
γ_1
EARTH

FIGURE 3


4000 PSIA
PRESSURANT TANK ASSEMBLY
MANUAL VALVE
FILTER
REGULATOR
RELIEF VALVE
PRESSURE CONTROL ASSEMBLY
CHECK VALVE
EXPLOSIVE VALVE
FILL AND VENT ASSEMBLY
PROPELLANT TANK ASSEMBLY
200 PSIA OPERATING PRESSURE
BURST DISK
PROPELLANT SUPPLY ASSEMBLY
TRIM ORIFICE
BIPROPELLANT VALVE
ENGINE AND HEAT SHIELD ASSEMBLY

Orbits Reference: Bate, Mueller, & White. Fundamentals of Astrodynamics. Dover Publications: New York. Copyright 1971.

Other Reference: Werts, Larson (editors). Space Mission Analysis and Design. Kluwer Academic Publications: Boston. Copyright 1991.

CHAPTER 3

ATTITUDE CONTROL

CHRIS STEWART

Chapter 3 - Attitude Control

Requirements

The attitude control subsystem must provide for correct orientation of the spacecraft so that all aspects of the mission can be carried out successfully. Pertinent aspects of the mission include transmitting and receiving spacecraft telemetry and ground commands, moderating thermal problems in transit to the moon, and landing the spacecraft softly, safely, and upright.

Components

Based in part on knowledge gained from the Surveyor and Viking landers, the requirements can be met by using a star tracker, a sun sensor, an inertial measurement system, a radar altimeter, a control computer, three-axis thrusters, and three landing thrusters. The star tracker and sun sensor data will be combined to provide reliable orientation upon arrival at the moon and to ensure that as the spacecraft spins while in transit, its spin axis is perpendicular to the sun's radiation vector, thus keeping the sides of the spacecraft at one moderated temperature and not heating one end or the other excessively. The inertial measurement system will receive orientation data from the star tracker and sun sensor upon arrival at the moon and then provide accurate attitude data to the landing thrusters to ensure proper landing attitude. The radar altimeter will measure height above the moon and descent rate to ensure a soft landing. The control computer will integrate all the sensors with the thrusters, and tie in the power system with the communications equipment and the attitude control equipment. The three-axis thrusters will set the initial spin on the spacecraft during transit and then orient the spacecraft properly to fire the landing thrusters. The landing thrusters will both slow the spacecraft to a soft landing and keep the spacecraft vertical with respect to the moon.

Specifications

The components used were chosen in accordance with the principle of keeping costs low by using off-the-shelf components. Tried and tested components also can be assumed to be reasonably reliable. The data shown in Figure 3-1 are for representative examples only, in order to get approximate values of size, weight, and power required.

Placement

The sun sensor will be placed on one side panel of the spacecraft, near the top. The star tracker, control computer, and inertial measurement unit will be placed on the top tier of the spacecraft. The radar altimeter attaches to the bottom, near but not too close to the landing thrusters, and two three axis thrusters will be attached parallel to each pertinent axis to ensure pure rotation about all axes.

Attitude Control Subsystem
Component Specifications

- Star Tracker: Ball OT-601, 8.7kg, 10W Power Required, Height: .295m, Diameter: .18m
  Ref: Corporate spec sheet
- Sun Sensor: Lockheed WASS, .92kg, .9W Power Required, Size: .0002 cubic meters
  Ref: Corporate spec sheet
- Inertial Measurement Unit: Gyros, 2.75kg, 10W Power Required. Ref: Wertz and Larson
- Radar Altimeter: Pulse type, 11.2kg, 24.9W Power Required, Transmit Freq.: 1 GHz, Range: 41.5m - 197km. Ref: Viking Lander
- Control Computer: Honeywell MILSTD1750A, 1MB Memory, 20W Power Required, 3.18kg, Size: .18m x .15m x .08m. Ref: Corporate spec sheet
- Three-axis Thrusters: Mono-prop Hydrazine, 4.45N Thrust, .1-.2kg each.
  Ref: Wertz and Larson
- Landing Thrusters: HS601 AKE (ARC/LPG), 1500N Thrust, 4kg each, Mono-prop Hydrazine
  Ref: Wertz and Larson

Figure 3-1

CHAPTER 4

COMMUNICATIONS

CHRIS STEWART

Chapter 4 - Communications

Requirements

The communications subsystem must provide reliable contact between ground stations and the spacecraft at all times, for several purposes. These purposes include sending spacecraft telemetry, receiving ground commands, and transmitting data from the scientific payload at a useful data rate.

Components

Based on a design requirement of data transfer at 1000 bits per second (BPS) at all times and the ability to receive all spacecraft communications on the ground with a 2 meter dish antenna, two types of spacecraft antennas were chosen. Data transfer will be accomplished with a high gain, .2 meter diameter parabolic dish antenna, and telemetry and commands will be relayed using an omnidirectional dipole antenna. A second, identical dipole antenna will be included to provide a backup. If the payload requires occasional data transfers at a higher bit rate, the control computer can store the data for a while and when the data is all collected, partially shut down the payload and allocate more power to the antenna to transfer the data at the higher bit rate.

Specifications

NASA reserves the frequency of 8.4 GHz for interplanetary communications, so that is the frequency the spacecraft will use. This dictates a dipole antenna of one-quarter the wavelength of the signal, or about 9mm. The dipole antenna will transmit and receive at a data rate of 100 BPS. The link budget, Figure 4-1, shows that with an assumed dipole antenna gain of 1.5, 39.5W of power will be needed to transmit spacecraft data once on the moon. That means that less than 40W will be needed to transmit telemetry while in transit to the moon. On the ground, a 2m parabolic dish with a gain of 15475 needs 13.7W to transmit at 100 BPS to the moon.

Payload data transfer at 1000 BPS will be accomplished using a .2m parabolic dish with a gain of 154.7 requiring approximately 4W of transmitted power to be received at the 2m dish on Earth.

Figure 4-2 shows the calculations used to arrive at these numbers and lists assumptions made.

Placement

The two dipole antennas will be placed at the ends of the spacecraft's legs, mounted on insulating material to prevent the heat of the moon's surface from conducting up the legs and degrading the antennas' performance. The parabolic dish will be placed on top of the spacecraft to allow for pointing accuracy and to keep the antenna cool and performing well, will be mounted opposite the RTG and as far away from the RTG's radiator as possible.

Figure 4-1

Link Budget

P_t - Transmitted Power
G_t - Gain of the transmitter
P_l - Path loss
G_r - Gain of the Receiver
P_r - Received Power
No - Noise Interference
BR - Bit Rate
E/No - Signal to Noise Ratio

Dipole Antenna
(Receiving Commands from Earth)

	Units(W)	dB
P_t	13.74648	11.38191646
G_t	15475.54	41.89645804
P_l	1.829e22	222.6230255
G_r	1.5	1.760912591
P_r	1.27e-18	-178.965655
No	5.52e-21	-202.580609
BR	100	20
E/No	15.8	11.98657087
Link Margin	2	3.010299957

Dipole Antenna
(Sending telemetry to earth)

	Units(W)	dB
P_t	39.58988	15.97584
G_t	1.5	1.760913
P_l	1.829e22	222.6230
G_r	15475.54	41.89646
P_r	1.27e-18	-178.966
No	1.59e-20	-197.987
BR	100	20
E/No	15.8	11.98657
Link Margin	2	3.010300

Parabolic Dish Antenna

	Units(W)	dB
P_t	3.837333891	5.840295894
G_t	154.7553970	21.89645804
P_l	1.829374e22	222.6230255
G_r	15475.53970	41.89645804
P_r	1.30915e-16	-158.830109
No	1.58976e-20	-197.986684
BR	1000	30
E/No	15.8	11.98657087
Link Margin	2	3.010299957

Figure 4-2

Assumptions

1. Noise Temperature of ground antenna (Ts_gr) is 1152 K

2. Noise Temperature of spacecraft antenna (Ts_s/c) is 400 K

3. Gain of Dipole antenna (G) is 1.5

4. Efficiencies (η) of both the parabolic antenna and the dipole antennas are .5

5. Length of dipole antennas is 1/4 wavelength

6. Link Margin is 2W

7. Signal to noise ratio (E/No) is 12.2

Equations and Data

Slant Range is mean Earth-moon distance and equals 384,400,000 m

Parabolic Antenna Gain:

$$G=\eta\left(\frac{\pi D}{\lambda}\right)^2$$

Path Loss (dB):

$$P_L=\left(\frac{4\pi R}{\lambda}\right)^2$$

Noise Interference:

$$No = k\ Ts$$

$$k = 1.38 \times 10^{-23}\ J/K$$

CHAPTER 5

THERMAL CONTROL

CHRIS STEWART

Chapter 5 - Thermal Control

Requirements

The spacecraft's temperature must be controlled so that all parts of the spacecraft stay within their respective temperature limits during all phases of the mission. The thermal design must do more than just allow all the components to survive and function, however; thermal control seeks to optimize the temperatures of all the spacecraft components so they all perform optimally.

Thermal Node Definitions

The following isothermal nodes will be defined on the LOLA spacecraft: top surface, each side face (6 total), the RTG and its radiator, the parabolic dish antenna, each leg (3 total) and each dipole antenna (2 total). This adds up to 14 thermal nodes whose temperatures need to be examined and perhaps controlled.

Nodal Shape/Configuration Factors

For the purpose of this section, it is assumed that all surfaces radiate and absorb as blackbody surfaces. The top surface of the spacecraft is a flat hexagon whose sides are 1.45m wide. Were the RTG and the parabolic dish antenna not present, the top would have a shape factor to space of one. However, the RTG and dish are very small compared to the area of the top surface (A_{top} = 5.256 m^2, A_{proj_RTG} = .262m^2, A_{proj_dish} = .03m^2), so a shape factor of only slightly less than one will be assumed. The dish antenna will usually see only space and itself, so if it is painted white or some other highly reflective color, most of its radiant energy goes to space, also. The RTG's radiator will radiate most of its energy to space due to its convex design, so a highly radiative surface will also be used there. The spacecraft's legs have a small enough surface area that radiation inputs will be negligible and conductance from the moon will dominate that problem. Each dipole antenna is extremely small and will be mounted on a reflective, insulating surface to keep it cool, so radiative interaction, and thus shape factors, are not of interest. Each face of the spacecraft sees both the moon and space, but very little else. The problem of defining a shape factor for one finite area (the spacecraft face) interacting exclusively with two other very large areas (space and the surface of the moon) encourages approximation of the areas of interaction involved instead of finding definite shape factors.

Joint Conductances

There must be very little conductance between the dipole antennas and the spacecraft legs so that when the lunar surface heats up, the legs do not transfer that heat to the antennas and reduce their efficiency. Mounting the antennas on an insulating surface will thermally isolate them, keeping conductive heat away.

Heat conducted between the spacecraft legs and body is not a serious concern because the spacecraft will not move once it lands on the moon. As long as the legs stay securely attached to the body, no thermal problems need to be examined. However, making the joints insulated would simplify the thermal problem for the body of the spacecraft, so if feasible, that will be done.

Among the sides of the spacecraft, internodal conductance is not much of a problem. As long as no one side gets above the design temperature, the rest of the sides are fine. Again, the sides do not need to move, so as long as they stay secure, the thermal problem is solved. They will be layered with thermal insulation (MLI) to insulate the inside of the spacecraft, and this will also keep internodal conductance very low.

The structure of the sides and top will both be supported by aluminum struts, and both will be stationary, so internodal conductance between any of the sides and the top is again of little significance.

The RTG will deploy to its horizontal position upon landing on the moon and then remain stationary. Once again, internodal conductance becomes largely insignificant, as long as the structure stays intact.

The parabolic dish antenna rests on a strut which has one hinge to allow the antenna to point continuously at Earth. This strut must have one constant temperature on both sides of the hinge, but the conductance between the top and the strut is only important in that the connection must remain secure.

In summary, each of the internodal cases has been examined, and because the spacecraft is stationary, internodal conductances are in most cases insignificant.

Heat Inputs

External:

Radiative inputs to the spacecraft include the sun at 1352 W/m^2, moon radiance at a high blackbody temperature of 380 deg. K and a low blackbody temperature of 120 deg. K (Ref: Book of Astronautical Constants), and albedo of the sun off the moon, which has an approximate hemispherical reflectance of .4. The earth, as viewed from the moon, subtends a solid angle of .000865 steradians. Therefore, the radiant inputs from the Earth are much too small affect the spacecraft. Heat will also be conducted into and out of the spacecraft by way of the legs resting on the surface as the surface heats up and cools down.

Internal:

The RTG will be used in conjunction with a variable conductance heat pipe to keep the spacecraft warm during the lunar night, and the radiator will dispose of excess heat during the lunar day (Figure 5-5 shows the necessary size of this radiator). The internal electrical components use minimal power and will have no significant heat input to the thermal situation.

Extreme Nodal Temperature Cases

The spacecraft will spin in transit to the moon with its spin axis perpendicular to the sun's radiance vector, so all the sides will be at a moderated temperature as compared to the temperatures they will experience when stationary on the moon. The maximum nodal temperature is calculated and shown in Figure 5-3 to be 467 degrees Kelvin on the top of the spacecraft during the lunar day. The minimum nodal temperature is theoretically zero for the top of the spacecraft because it has no radiative inputs from space, but the presence of the RTG actually radiates a little to the top, keeping it above zero. These extreme temperatures are calculated assuming perfect, blackbody interaction.

To moderate these temperatures, several steps are taken. The first is to apply a coating to the sides and top of the spacecraft to reduce those nodes' solar absorptance and thus reduce solar heat input. Figure 5-4 details this, and shows that white paint with a solar absorptance of .2 and an infrared emissivity of .9 will cool the top of the spacecraft to approximately 320 degrees Kelvin. The second step is to attach a variable conductance heat pipe to the hot junction of the RTG and to part of the interior of the spacecraft so that as the outside temperature decreases, more heat will spread into the spacecraft, keeping the payload and electronics at a moderate temperature. A flexible joint must exist in the heat pipe so that the RTG may swivel from its vertical launch, transit, and landing position to its horizontal stationary position. This poses greater engineering challenges, but it has been done before with some success. (For details on the operation of heat pipes, see Space Mission Analysis and Design by Wertz and Larson.) The interior temperature may be moderated by painting the interior surfaces of the spacecraft black, effectively creating one blackbody radiator so that all the areas in the interior at one temperature. Thermal insulation (MLI) on the sides of the spacecraft will keep the heat from the heat pipe from immediately radiating into space and allow components to stay warm.

FIGURE 5-1

SIDE OF S/C

Area of moon's surface the spacecraft can "see":

Assuming an average height of 1 m from the surface:

RADIANT MOON INPUT

$$dE = \frac{ds_i \, \sigma T^4 \cos\phi_1 \, ds_2 \cos\phi_2}{\pi \, r_{1-2}^2},$$

r_{1-2} is distance from lunar surface to s/c

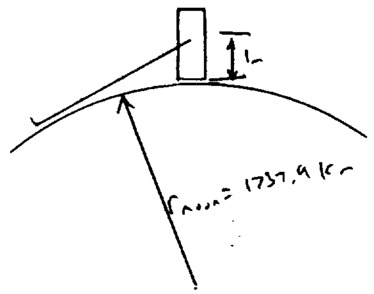
ds_i : is differential area of lunar surface

ds_2 is differential area of S/C side

$\cos\phi_1$ is angle between surface normal and r_{1-2}

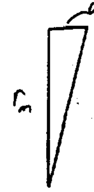
$\cos\phi_2$ is angle between S/C side and r_{1-2}

T^4 is black body temp of moon (396°K)



r_{1-2} maximum is 1.864 km., approximate and use 1/3 of that

$ds_{i_{max}}$ πd_{max}^2, approximate with $\pi\left(\frac{d_{max}}{3}\right)^2 = 1.213\times10^6 m^2$



$d_{max} = \sqrt{\ell^2 - r^2} = 1.864$ km

$\ell = (1737.9 + .001)$ km

$(\cos\phi_1)_{max}$ is $\cos(90 - .0307)$, approximate with $\cos\left(90 - \tan^{-1}\left(\frac{1}{1864/3}\right)\right)$, or .0016

$(\cos\phi_2)_{max}$ is $\cos(90 - \tan^{-1}(1864))$, approximate with $\cos\left(90 - \tan^{-1}\left(\frac{1864}{3}\right)\right) = .99\bar{9} \approx 1$

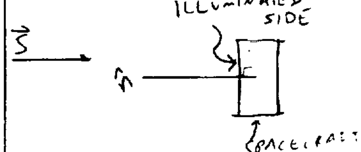
$ds_2 = (1.45m)(1.5m) = 2.175 \, m^2$

$T^4 = 2.459\times10^{10} \, K^4$

$$dE = \left(1.213\times10^6 m^2\right)\left(5.67\times10^{-8}\frac{W}{m^2 \, {}^\circ K^4}\right)\left(.0016\right)\left(2.175 m^2\right)\left(2.459\times10^{10} K^4\right) \Big/ \left(\frac{1864 m}{3}\right)^2 \cdot \pi$$

$dE = 4.852$ W ⇐ MOON RADIANT INPUT TO A SIDE OF THE SPACECRAFT

For Sun input, $q_{in_{sun}} = \int_0^{90} S A \cos\theta \, d\theta$, θ = angle between $\vec{S}$ and $\hat{n}$, CHANGES AS SUN MOVES



$S = 1352 \, W/m^2$

$q_{in_{sun}} = 1352 \frac{W}{m^2} \cdot (1.45m)(1.5m) \int_0^{90} \cos\theta \, d\theta$

$q_{in_{sun}} = 2940 \sin\theta \Big|_0^{90} = 2940$ W

$q_{in_{sun}} = 2940$ W ⇐ heat input to SIDE OF S/C BY THE SUN

ORIGINAL PAGE IS OF POOR QUALITY

MOON REFLECTION (ALBEDO) INPUT:

$$q_{ALBEDO} = q_{into\ moon} \cdot a \frac{1}{\pi} dS_{face} \cdot \cos\phi_{face}\ dS_{moon} \cos\phi_{moon\ \perp} \Big/ r^2_{moon-face}$$

$$q_{into\ moon} = 1353 \frac{W}{m^2} \Leftarrow \text{SOLAR RADIANCE}$$

(FROM RADIANT MOON INPUT)

a: hemispherical reflectance $\simeq .4$

$dS_{face} = 2.175\ m^2$

$\cos\phi_{face} \cong 1$

$dS_{moon} \cong 1.213 \times 10^6\ m^2$

$\cos\phi_{moon} \simeq .0016$

$r_{moon-face} \cong 621.3\ m$

$$q_{ALBEDO} = 1353 \frac{W}{\cancel{m^2}} \cdot .4 \cdot 2.175\cancel{m^2} \cdot 1 \cdot 1.213 \times 10^6 \cancel{m^2} \cdot .0016 \Big/ \left((621.3\cancel{m})^2 \pi\right)$$

$$q_{albedo} = 1.884\ W$$

DURING THE LUNAR NIGHT, $T_{moon} = 120°K$

SO RADIANT MOON INPUT (NIGHT) WILL BE:

$$q_{moon\ night} = \left(1.213 \times 10^6 \cancel{m^2}\right)\left(5.67 \times 10^{-8} \frac{W}{\cancel{m^2}\,°K^4}\right) \cdot (120°K)^4 \cdot (2.175\cancel{m^2})(.0016)(1) \Big/ \pi \cdot (621.3\,m)^2$$

$$q_{moon\ night} = .0409\ W$$

FIGURE 5-3

IF α & ϵ OF SIDES OF S/C ARE EQUAL TO 1, ITS TEMP WILL BE: (HOT CASE)

$$A_{S/C\ SIDE}\,\epsilon\,\sigma T^4 = 4.852\ W + 2940\ W + 1884\ W = q_{in} = q_{out} = 2947\ W$$

$$T = \sqrt[4]{\frac{q_{in}}{A_{S/C\ SIDE}\,\epsilon\sigma}} = \sqrt[4]{\frac{2947\ W}{(2.175\,m^2)(1)(5.67\times10^{-8}\ \frac{W}{m^2\,°K^4})}} = 393.2\ °K = T_{HOT\ SIDE}$$

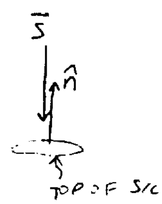
(COLD CASE:)

$$T = \sqrt[4]{\frac{q_{moon\ night}}{A_{SIDE}\ \epsilon\sigma}} = \sqrt[4]{\frac{.0404\ W}{(2.175\,m^2)(1)(5.67\times10^{-8}\ \frac{W}{m^2\,°K^4})}} = 24.0\ °K$$

TOP OF S/C: (α AND ϵ ARE 1)

COLD CASE: THEORETICALLY, SINCE IT DOESN'T SEE THE MOON, THE TOP OF THE S/C COULD DROP TO 0 °K DURING THE LUNAR NIGHT.

HOT CASE:



$$q_{in} = \int_{-90°}^{90°} S \cdot A_{top} \cos\theta \; d\theta = (5.256\,m^2)(1353\ W/m^2)\ \sin\theta \Big|_{-90}^{90} = 14222.7\ W$$

$$A_{top} = (1.45\,m)(2.9\,m) + (.725\,m)(1.45\,m) = 5.256\,m^2$$

$$q_{in} = 14227.7\ W$$

$$T = \sqrt[4]{\frac{q_{in}}{A_{top}\,\epsilon\sigma}} = \sqrt[4]{\frac{14227.7\ W}{(5.256\,m^2)(1)(5.67\times10^{-8}\ \frac{W}{m^2\,°K^4})}} = 467\ °K$$

$T_{TOP} = 467\ °K$ $\Leftarrow$ MAXIMUM HOT CASE

MODERATING THE SIDES

TO ENSURE A MAXIMUM TEMPERATURE ON A SIDE OF LESS THAN 50° C (323 °K); USE PAINT WITH α LESS THAN 1 AND ϵ CLOSE TO 1:

WITH WHITE PAINT ($\alpha = .2$, $\epsilon = .9$):

$$A_{\substack{S/C \\ SIDE}} \epsilon \sigma T^4 = q_{in} \cdot \alpha = (2947 W)(.2) = 589 W$$

$$T = \sqrt[4]{\frac{589 W}{(2.175 m^2)(.9)(5.67 \times 10^{-8} \frac{W}{m^2 °K^4})}} = 269 °K = T_{\substack{side \\ max}}$$

RAISING α SLIGHTLY CAN BRING $T_{\substack{SIDE \\ max}}$ UP TO AROUND 323 °K,

AND USING GREYER PAINT WILL RAISE α.

RAISING MINIMUM TEMPERATURE - HEAT PIPE

MODERATING THE TOP

AGAIN, SO THAT $T_{MAX} < 50°C$, USE WHITE PAINT ($\alpha = .2$, $\epsilon = .9$):

$$A_{\substack{S/C \\ TOP}} \epsilon \sigma T^4 = q_{in} \alpha = (14227.7 W)(.2) = 2845.5 W$$

$$T = \sqrt[4]{\frac{2845 W}{(5.256 m^2)(.9)\ 5.67 \times 10^{-8} W/m^2 °K^4}} = 320.9 °K = T_{\substack{max \\ TOP}}$$

THIS PUTS $T_{\substack{MAX \\ TOP}}$ RIGHT WHERE REQUIREMENTS SPECIFY.

RTG HEAT

TO DISSIPATE 3500 W OF POWER AT 30°C ; WHICH ARE THE CONDITIONS AT WHICH THE RTG'S COOLING SYSTEM OPERATES; A RADIATOR IS NEEDED (ASSUME WHITE PAINT WITH $\epsilon = .9$):

$$q = A_{RAD} \epsilon \sigma T^4$$

$$3500\,W = A \cdot .9 \cdot 5.67 \times 10^{-8} \frac{W}{m^2 K^4} \left((30 + 273)K\right)^4$$

$$A_{RAD} = \frac{3500\,W}{(.9)(5.67 \times 10^{-8} \frac{W}{m^2 K^4}) \cdot (8.43 \times 10^9\, K^4)} = 8.137\, m^2$$

IF THIS RADIATOR IS IN THE SHAPE OF A SIMPLE FLAT CIRCLE, ITS RADIUS WOULD BE

$$A_{rad} = \pi r^2$$

$$r = \sqrt{\frac{A_{rad}}{\pi}} = \sqrt{\frac{8.137\, m^2}{\pi}} = 1.61\, m$$

THIS CIRCLE MAY BE FOLDED UP LIKE A FAN AND DEPLOYED AFTER LANDING ON THE MOON.

CHAPTER 6

POWER SYSTEMS

CHARLES TURNER

CHAPTER 6 - POWER SYSTEMS

Requirements

For LOLA, a power source is needed to provide 135 watts constantly both day and night - 100 watts for the experiment and 35 watts for communications and other various components. The power source needs to be reliable and be able to fit into the mission weight constraints.

Choice of Power Source

For the power system of LOLA, the general purpose heat source radioisotope thermal generator (GPHS-RTG) was chosen as the power source. Batteries were considered as a power source at first, but due to the mission's weight requirements, calculations proved them to be too bulky to be practical. The GPHS-RTGs provides full power during both night and day, supplies enough excess heat to warm LOLA's equipment during the nighttime, and has a relatively low weight. The information for the GPHS-RTG was taken from the Final Safety Analysis Report for the Ulysses Mission, published for the United States Department of Energy by General Electric's Astro Space Division in March 1990 and provided courtesy of Mr. Wayne Brittain, Program Manager for Advanced Nuclear Systems at Teledyne Energy Systems, Timonium, MD. Diagrams of the GPHS-RTG are included in Figures 6-1 and 6-2.

Power Source Specifications

The initial specifications for the GPHS-RTG are shown in

Table 6-1. Since LOLA needs only 135 watts, the specifications listed in Table 6-1 can be scaled down to meet the mission needs, according to Mr. Brittain, and are listed in Table 6-2. The operational life of the GPHS-RTG is 4.65 years, which is considerably longer than the 3-year mission life.

<u>GPHS-RTG Fuel</u>

The fuel for the GPHS-RTG is a mixture of plutonium in the form of PuO_2 with 85% of the fuel being $^{238}PuO_2$. Each of the 72 pellets are encapsulated and sealed hermetically in such a way to prevent radiation emission and to protect the other components from thermal damage. No mention is made in the report if any emission of radiation has actually been measured, something that would be of concern especially if there is a launch abort or if humans want to approach the lander when it is on the moon.

<u>Thermal Control</u>

The GPHS-RTG has internal heat control systems already built in. A gas management system using an inert gas atmosphere controls the heat inside the GPHS-RTG and protects the internal workings and refractory of the GPHS-RTG while the unit is on the ground (30 days maximum) and in storage (3 years maximum). Once in space, a pressure relief device vents the gas atmosphere, allowing full power operation.

While in space, the heat is dissipated through cooling passages that are part of eight tapered fins. The fins are

E-beam welded to the longitudinal ribs and stub fins of the unit housing. An active cooling system is used in the cooling passages to dissipate about 3500 watts with a inlet coolant temperature of 30°C and an average radiant cooling sink temperature of 25°C. A silicone base paint made by General Electric having a minimum emissivity of 0.90 is applied to the outer housing assembly to help speed up the heat transfer.

The GPHS-RTG is placed on top of LOLA and swivel off to the side slightly when the lander touches the lunar surface to make room for the parabolic antenna. It is separated from the rest of the lander by radiative material that will shield the excess heat of the GPHS-RTG from the lander's equipment. The radiative material is painted white and has a high emissivity to aid in dissipating heat. The heat that is needed to warm LOLA's equipment during the nighttime is brought from the GPHS-RTG by a special conductive heat pipe.

Launch Failure Hardening

The GPHS-RTG has been designed to withstand impact if the launch platform should fail and the payload reenters the atmosphere. The GPHS-RTG has two Graphite Impact Shells to provide impact protection and also has aeroshells, which also serve as the primary structural members of the GPHS, to act as protection for the Graphite Impact Shells from the aerothermodynamic environment that may be encountered during reentry.

Cost

According to Mr. Brittain, it is hard to determine a precise cost figure for the GPHS-RTG mainly because of the procurement process for the nuclear fuel. The Department of Energy controls all sales of plutonium in the United States, and the price of plutonium is negotiated at every sale of the fuel. He says that $5 million is a good estimate for this project.

Overview

The GPHS-RTG is most reliable choice for the power source for the LOLA mission. Taking up minimal weight, it provides the necessary power at all times while also providing heat at night. Although the excess heat is a problem during the daytime, it is solved by separating the GPHS-RTG from the rest of the lander by a radiative material. The GPHS-RTG is definitely the best choice for the power source for this mission.

DESIGN SPECIFICATIONS FOR THE GPHS-RTG

RTG Electrical Power Output:	285 W BOL 254 W EOL
Operation Life:	40,000 hours after launch 4.56 years
Weight:	123.5 pounds (56.14 kg)
Output Voltage:	28-30 volts
Envelope:	about 18 in. (0.46 m) diameter by about 45 in. (1.14 m) long
Hot Junction Temperature:	1000°C
Fuel:	PPO (84% $^{238}PuO_2$)
Thermoelectric Material:	SiGe
Magnetic Field:	about 30 nano tesla (30 X 10^{-5} Gauss) at 1 meter
On pad:	30 days unattended capability
Storage life:	3 year ground storage
Auxiliary cooling:	Remove $\geq$ 3500 watts with an average sink temp. of 25°C and an inlet coolant temp. of 30°C

Table 6-1

GPHS SPECIFICATIONS FOR LOLA

Power Output:	151.5 W BOL, 135 W EOL
Operational Life:	40,000 hours after launch or 4.56 years
Weight:	65.64 pounds or 29.84 kg
Output Voltage:	28-30 Volts
Envelope:	0.46 m diameter by 0.57 m long
Hot Junction Temperature:	1000° C
On Pad:	30 day unattended capability
Storage Life:	3 year ground storage
Auxiliary Cooling:	Remove $\geq$ 3500 watts with an average sink temp. of 25°C and an inlet coolant temp. of 30°C

Table 6-2

Figure 6-1

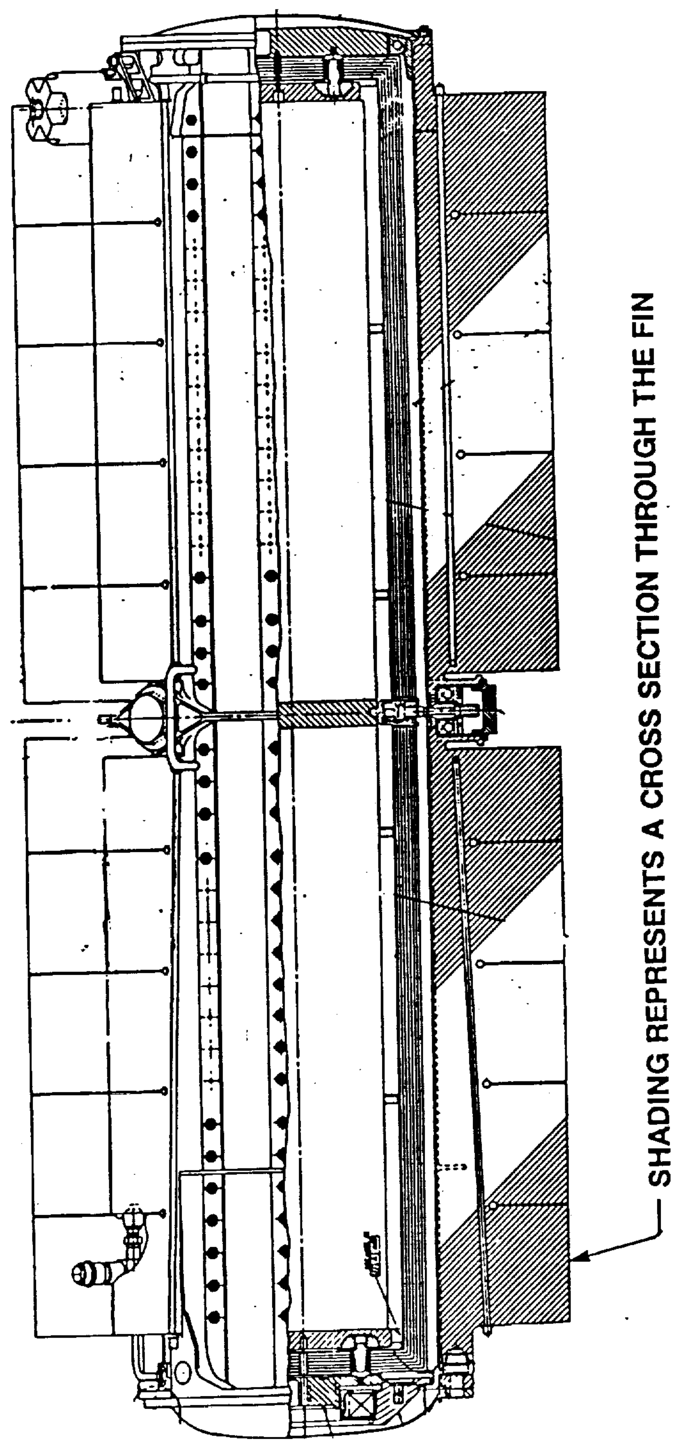


Figure 1-14. GPHS-RTG Assembly Profile

Figure 6-2

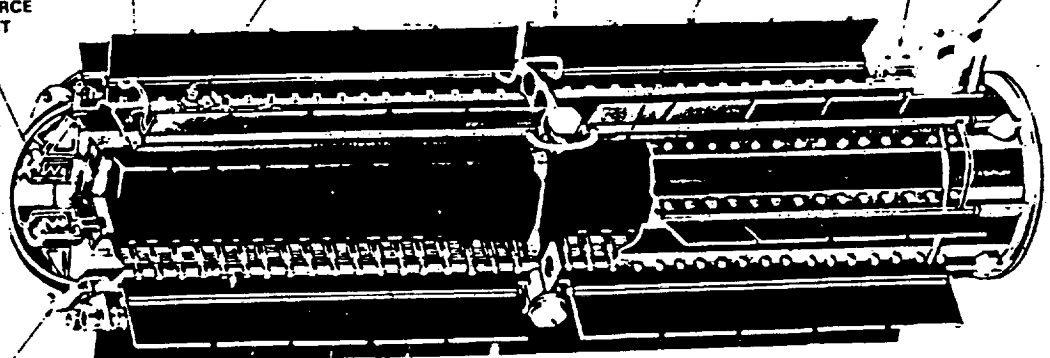


Figure 1-1. GPHS-RTG

CHAPTER 7

STRUCTURES

GAVINO RIVAS

Chapter 7 - Structure and System Integration Design

Purpose:

The lander (Fig 7-1) structure had several missions to accomplish during its operation to the moon. Several of these duties include:

- Protective structure for fairing retention
- Support structure for mounting both control and experimental apparatus
- Protective structure for transit to the moon and landing
- Base of operations during experimental phase
- Basic thermal control

Materials:

The structure itself is made of high grade aluminum to reduce weight with out reducing strength. This material has already proven itself on the lunar surface during such space missions as the Surveyor program. It is also cost effective, with a price tag much lower than other composite materials of similar strength and weight. The frame will consist mostly of metal tubing with honey comb plates serving as the mounting bases for the experimental and support packages. The sides of the craft will consist of basic thermal blankets draped between the main support members.(fig 7-1) This was done to provide for basic thermal control for the lander.

Structure:

The structure chosen by the LOLA team was a hexagonal cylinder which was fully supported by a truss frame in the axial direction.(fig 7-2) This was the direction from which all of the force was expected to occur. The structure had two tiers, one for the support equipment and one for the experiment itself. The team tried to maximize the area on and over the experimental bay, thus allowing for most any dimension of experimental package. Three access hatches to the moon's surface were placed flush with the bottom of the lander to allow for maximum access to the lunar surface. With three openings, the maximum amount of access to the outside can be allowed. The structure must provide a base of operations for the experimental package for up to three years after its deployment so it must be able to with stand the landing with out any major damage.

Power:

Power from the RTG needed to be distributed throughout the lander. There was a transformer which receives the power from the RTG each of the sub systems and experiment package connect to it.(fig 7-3) The RTG was configured to provide enough power to all of the sensors and subsystems and also provide 100 watts of power to the experiment as well. The experimental feed could be located where ever is needed by the experiments designers.

Communications:

The communications package involves two separate systems.(fig 7-4) First there are two sets of dipoles located on two of the legs. These dipoles are for attitude adjustments and other necessary contacts between the lander and mission control. These omni-directional antennas provide a redundant manner in which the space craft could be controlled. The second is a .5 meter dish, used for data transfer for the experiment. The nominal transfer rate is 1000 bits per second. The communication package includes a transceiver which is connected to the on-board computer.

Sensors:

The lander contains three sensors, two for attitude control and one for landing.(fig 7-5) The two for attitude control include a star sensor and a sun sensor with the star sensor located along side the rest of the support equipment and the sun sensor located just below the top tier. Both would have openings for the optical portion of the sensor but the rest would be covered buy the blanket. A radar altimeter was located at the bottom of the craft, in between the fuel tanks and landing thrusters. This would be used for landing only and, like the other two sensors, would then be shut off after used.

Thrust:

For landing, the space craft will have three thrusters located under the bottom tier.(fig 7-6) These thrusters will be connected to actuators controlled bye the on-board computer. Instead of having two different systems to control the landers decent and attitude, these can be adjusted to control both rate of decent and attitude. The tanks for the landing system, as well as the actual thrusters, will take up about .4 meters of the space craft.

Launch:

The craft would be folded (fig 7-7) in the fairing itself and open once deployed. The legs would fold flush to the sides of the lander. The dish would collapse on itself and rotate down to protect itself. The RTG would be rotated up to allow all of the force of its weight to be transmitted along the axis of the space craft. This would also protect the RTG upon impact with the lunar surface. Neither it nor the dish will be deployed until the lander has reached the lunar surface. The support structure is necessary to hold the lander in place during launch.

Payload:

The platform upon which the experiment would be attached would be a hexagon, slightly smaller than the landers actual diameter.(fig 7-8) The members required for the structural integrity of the craft must be avoided as well as those required to prevent buckling upon impact with the lunar surface. The center of the craft is open for the experimental package and there is also areas open for access to the doors and the moon itself.

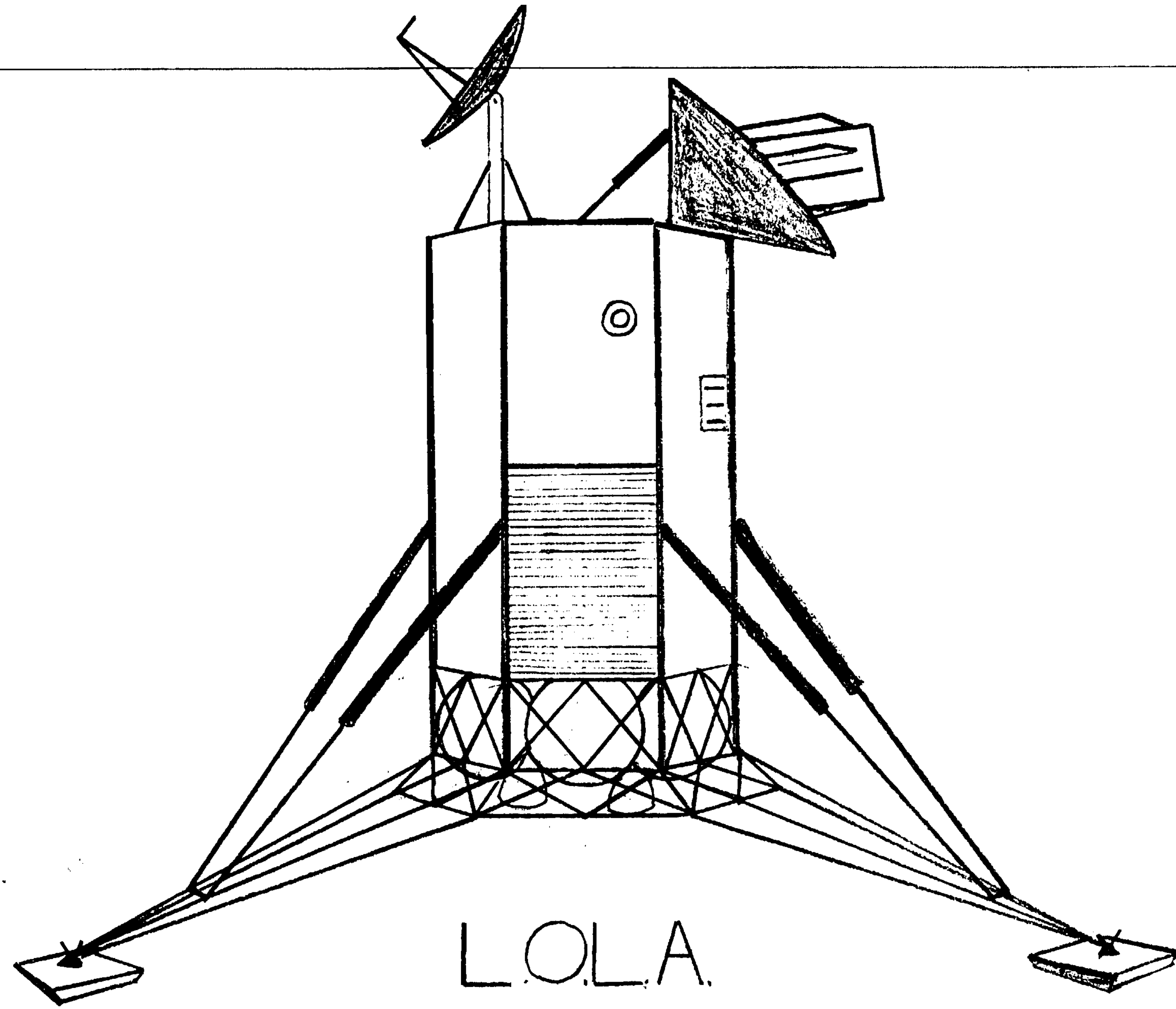
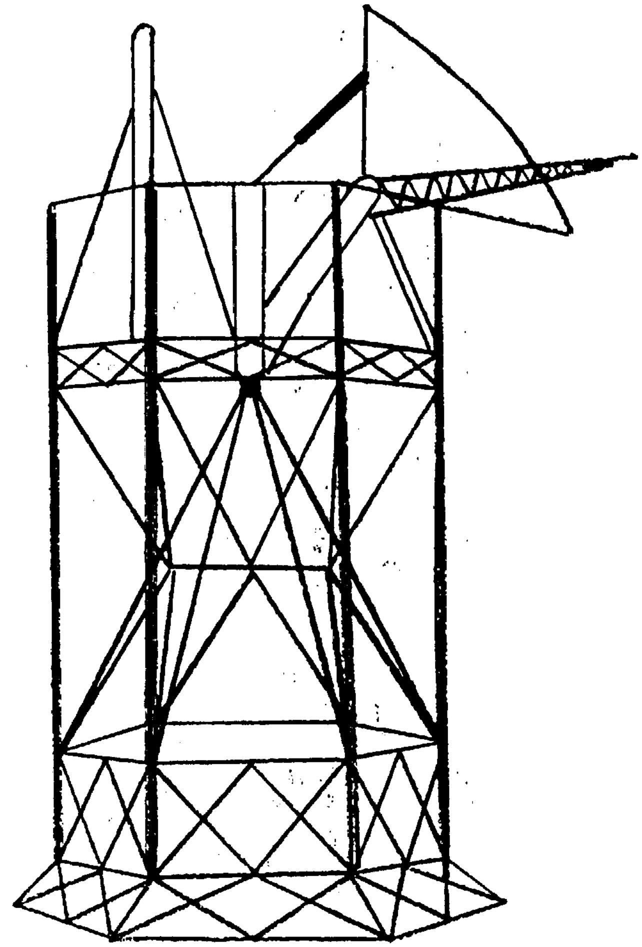
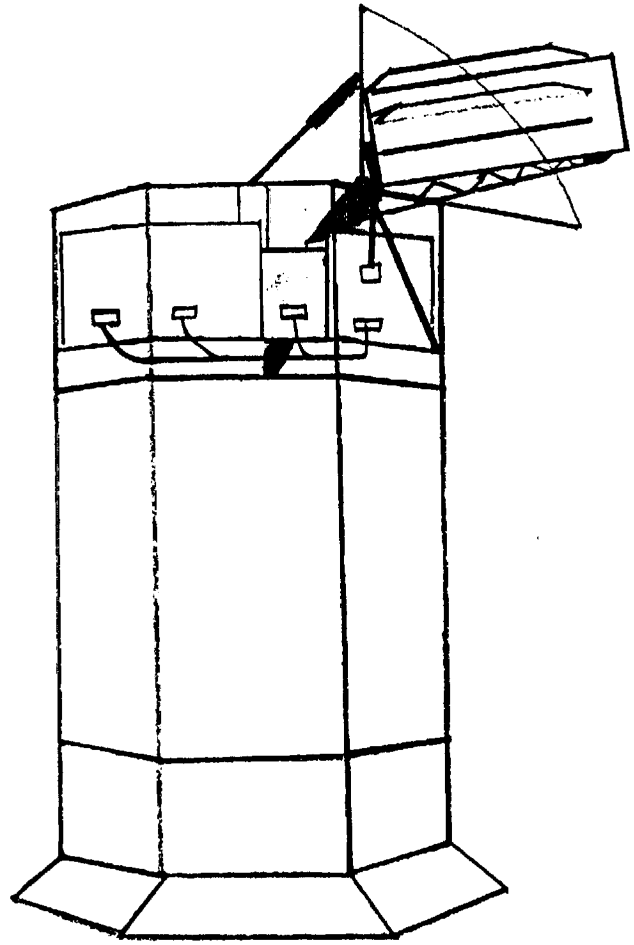


fig 7-1



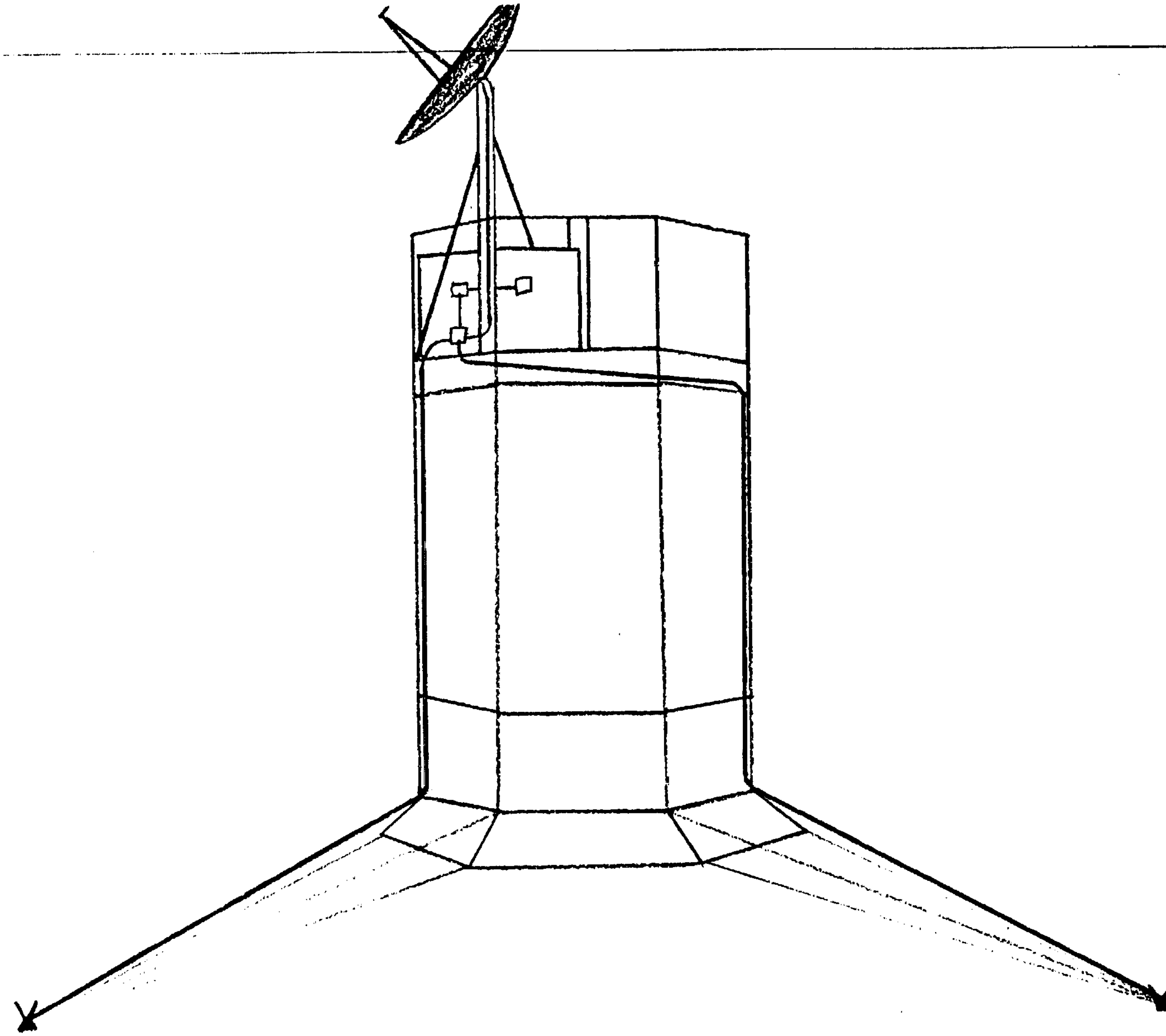
STRUCTURES

fig 7-2



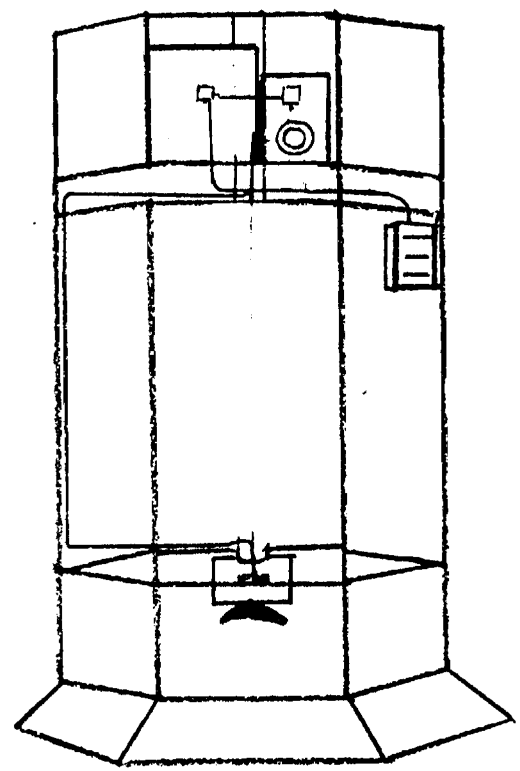
POWER

fig 7-3



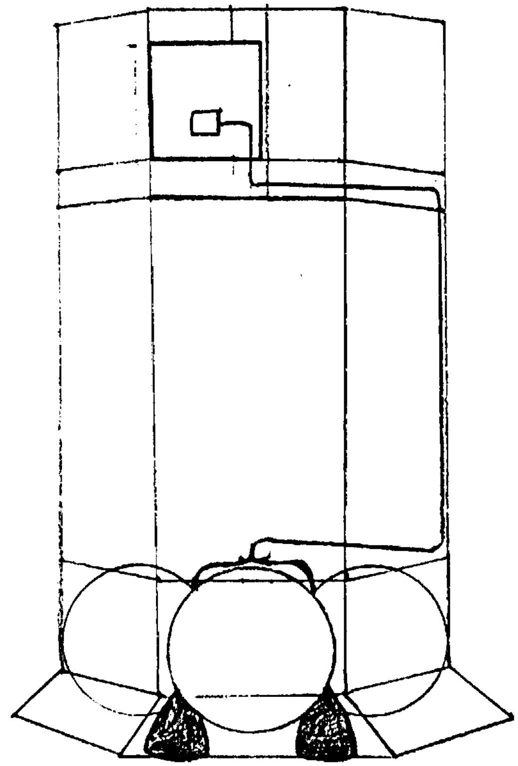
COMMUNICATIONS

fig 7-4



SENSORS

fig 7-5



THRUST

fig 7-6

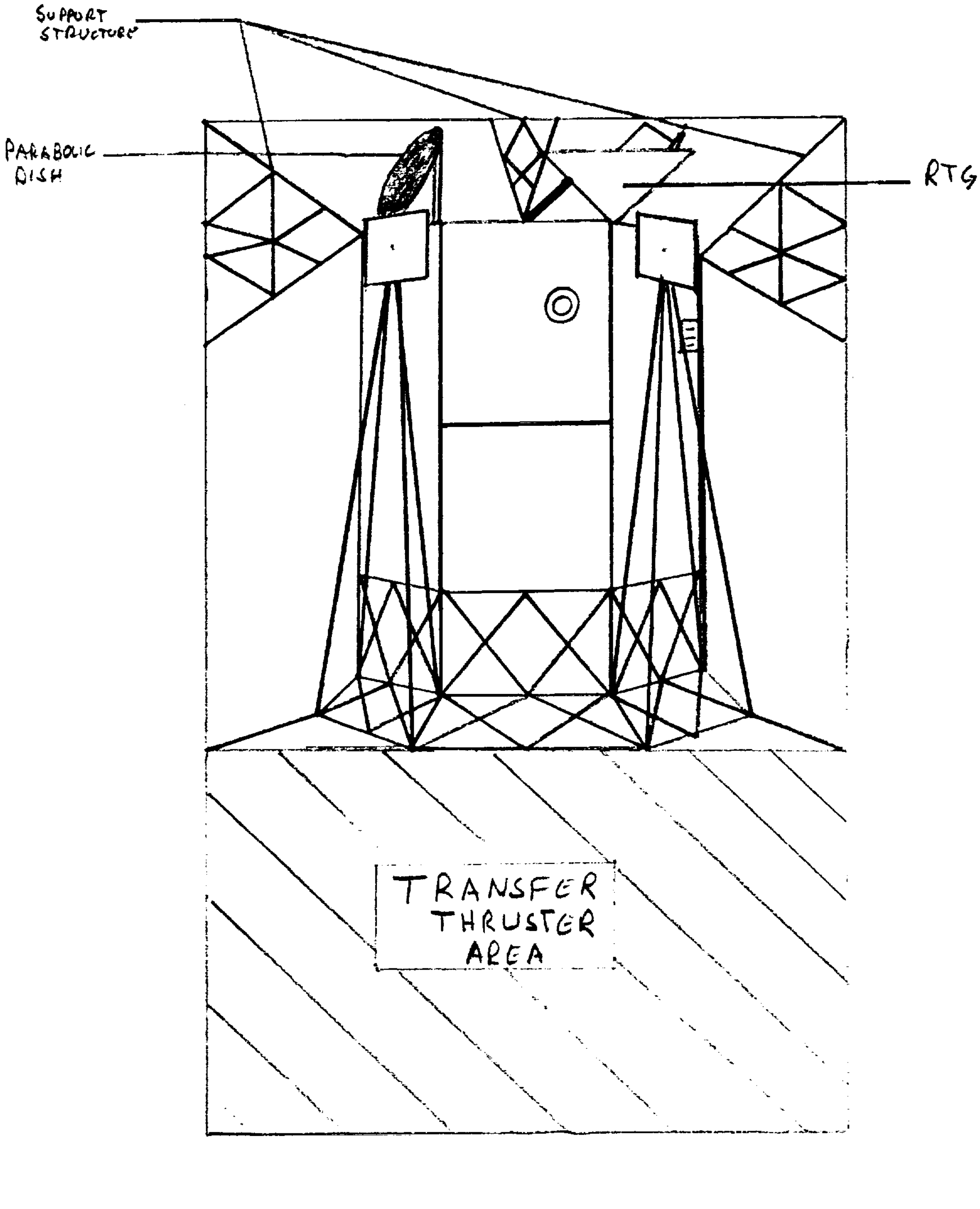

SUPPORT STRUCTURE
PARABOLIC DISH
RTG
TRANSFER THRUSTER AREA

Fig 7-7

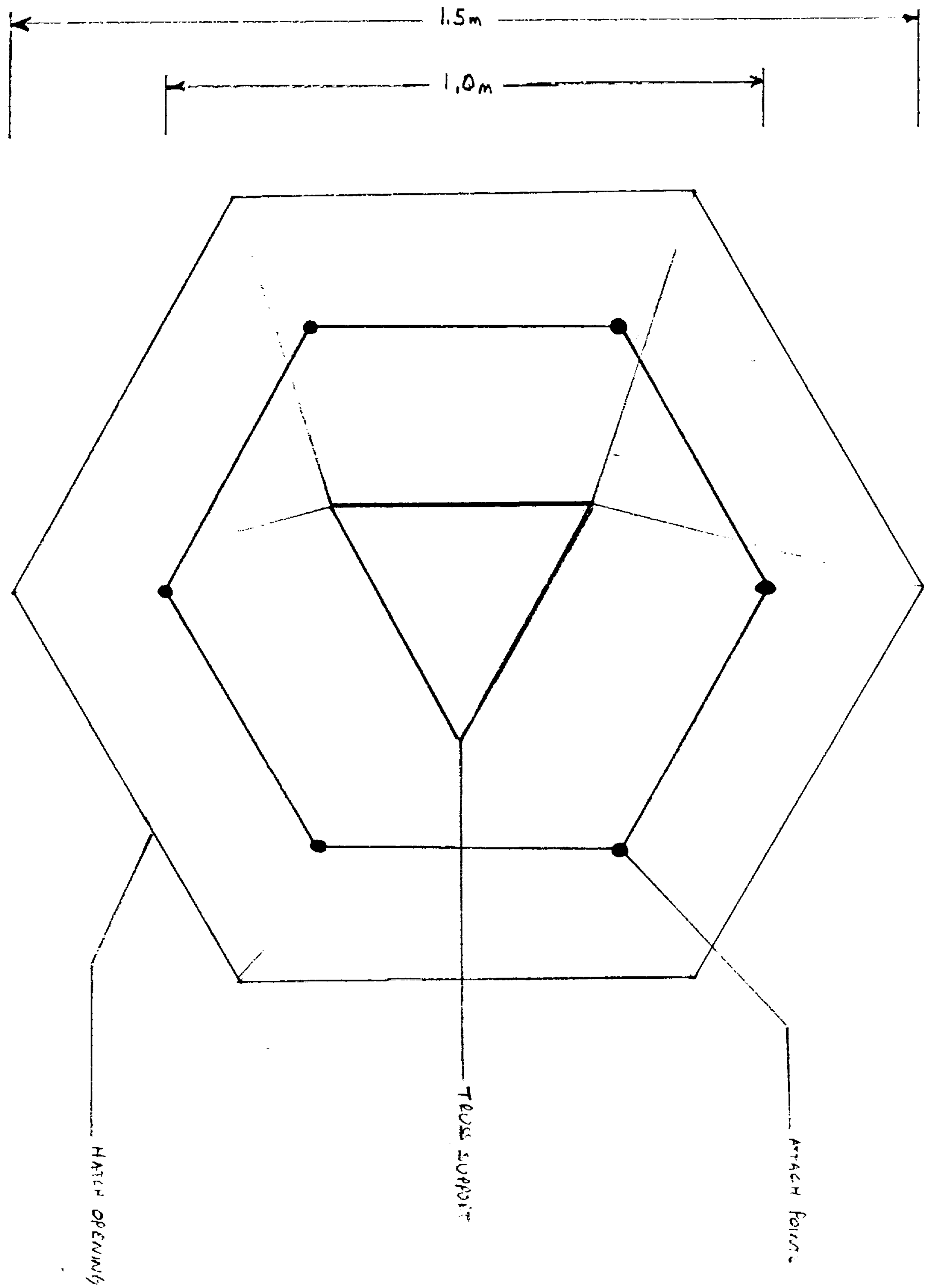


Fig 7-8

CHAPTER 8

CONCLUSION

MIKE ABREU

Chapter 8 - Conclusion

The Lunar Operations Landing Assembly design has shown itself to be a possible valuable asset to the Space Exploration Initiative. Using the components discussed in the preceding chapters, the lander will be able to be built from off-the-shelf components, thus holding down costs and increasing the probability of good mission reliability. A few key design subsystems, however, call for further investigation and analysis.

One of the design areas that warrants further investigation is thermal control of the spacecraft and payload bay area. The use of variable conductance heat pipes will help immensely in this effort, but detailed analysis is necessary in order to determine potential stopgaps. The thermal control of the payload bay will be of considerable challenge, but if it is met with success, it may be of great value to future lunar and Mars lander missions.

Another area of concern is the use of the radio-isotope thermal generator. This mission will be the first use of some form of nuclear power on another heavenly body's surface. Previously, RTG's have been used on planetary explorers such as Ulysses and Galileo, but not on any type of lander. Also, RTG's will play a significant role in the public attention directed towards this mission, as any form of nuclear power in space must be seen as reliable and safe by the American public and environmentalists. The performance, safety, reliability, and cost of RTG's on lunar landing missions must be evaluated thoroughly.

A complete cost analysis study is beyond the scope of this report, but it is essential to the furtherance of the lunar lander mission. A low-cost spacecraft will provide a means to explore the lunar surface as often as possible, thereby facilitating future missions and the possible selection of a lunar base site. Preliminary cost estimates for the Delta II launch vehicle, RTG, and other components is expected to be under $100 million. Since cost estimates are rarely stable, a detailed cost analysis study is crucial to the completion of the mission. In regards to future manned and unmanned exploration of space, costs may very well be reduced by joint cooperation with other nations. LOLA provides an efficient means to unify international support for good scientific studies of the Moon in furtherance of SEI objectives.

The LOLA design is one approach to the problem of landing scientific payloads on the lunar surface. Extensive collaboration on the part of the design team facilitated the design of choice, and provided an opportunity to overcome some of the obstacles presented by such a constrained mission. The future of manned lunar and martian ventures begins here, with a first effort back to the lunar surface as a stepping stone to the stars.